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THE NEED FOR NEW MATERIALS IN AGING AIRCRAFT STRUCTURES

R.J. Bucci and C.J. Warren
Aluminum Company of America
Alcoa Center, PA 15069 USA

E.A. Starke, Jr.
Department of Materials Science and Engineering
University of Virginia
Charlottesville, VA 22903 USA

SUMMARY

The end of the Cold War and political and economic considerations has resulted in an effort to extend the life of many aircraft that are the backbone of NATO operational forces. Although some are designated to be replaced with new aircraft, the replacement schedule of many often requires an unprecedented life span of between 40 to 60 years before retirement. Many of the older aircraft have encountered, or can be expected to encounter, aging problems such as fatigue cracking, stress corrosion cracking, corrosion and wear. In order to ensure continued airworthiness and flight safety the structural components undergoing these problems will have to be repaired or replaced. Alloy development that has taken place since a large percentage of the older aircraft were put into service has resulted in several new materials, heat treatments and processing technology that can be used for appropriate longer lasting and higher performing airframe components thus reducing life-cycle costs. This paper describes some of these materials and their advantages over those suffering from "aging problems."

1. INTRODUCTION

In 1996 the U.S. Air Force requested the National Research Council (NRC) to identify research and development needs and opportunities to support the continued operations of their aging aircraft. The results of this study, which was undertaken by a committee selected by the National Materials Advisory Board of the NRC, were published in the Committee's final report in 1997.[1] Among the many recommendations made by the Committee, one was to develop guidelines to broaden the application of improved materials as substitutes for incumbents with low damage tolerance and corrosion resistance. Such substitutions must make good business sense with respect to reduction in life-cycle costs and materials availability. Examples of reducing the life-cycle cost by implementing new materials on aging aircraft structure are given in the paper by Austin et al in this proceedings.[2]

The U.S. Air Force, as well as the air forces of other NATO countries, has many old aircraft that form the backbone of the total operational force structure. Many of these, e.g. the KC-135, the B-52, and the C-141were introduced into service in the 1950s and 1960s. Even the F-15 air superiority fighter became operational 20 to 25 years ago and the F-16 and KC-10 jet trainer at least 15 years ago. The extended use of these aircraft,

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in conjunction with frequent changes in mission requirements, results in increased maintenance and repair costs associated with structural cracking and corrosion problems which are, in most cases, associated with aluminum alloys and tempers developed prior to 1960.

Structural (fatigue) cracking is a direct result of aircraft use; i.e. load or stress cycles, and will eventually occur in all aircraft. Corrosion results from the exposure of susceptible materials to various corrosive environments, e.g. humid air, saltwater, sump tank water and latrine leakage, and to inadequate or deterioration of corrosion protection systems. In the case of aluminum primary structure, numerous service difficulties have been documented on components manufactured from alloys 2024-T3, 7075-T6, 7178-T6 and 7079-T6. For example, in order to minimize structural weight and thus maximize payload capability of the KC-135, the Air Force elected to use 7178-T6 in the lower wing skins and 7075-T6 in other locations in the aircraft. These alloys were designed to emphasize strength and have low damage tolerance. In 1977 the Air Force recommended that the wings be redesigned using more damage tolerant 2024-T3 and also recommended cold working fastener holes in the remaining 7178-T6. However, there is currently concern about the long-term effectiveness of the cold worked fastener holes and structural deterioration of the 2024-T3 due to exfoliation corrosion and multi-site corrosion-fatigue damage.[1]

Research since 1960 has led to the development of several new aluminum alloys, heat treatments and processing methods that offer more damage tolerant and corrosion resistant alternatives for airframe components than those that were used in the older aircraft. The overaged T73 and T76 tempers were developed in the early 1960s to make 7075 more corrosion resistant to stress corrosion cracking and to exfoliation corrosion; however, the improvement obtained is at the expense of strength. In 1974 Cina obtained a patent[3] specifically targeted at 7075, for a heat treatment procedure to provide stress corrosion resistance equivalent to an overaged T73 temper while maintaining the peak-aged strength. Although the concept, called retrogression and re-aging (RRA), seemed industrially impractical at the time, derivative tempers have been taken to practice as will be discussed in the paper by Holt et al in this proceedings.[4] In the 1970s alloy 7050-T74 was developed to fill the need for a material that would develop high strength in thick section products, good resistance to exfoliation corrosion and stress corrosion cracking, and adequate fracture toughness and fatigue characteristics. Also, in the 1970s a derivative of 7075, i.e. 7475, was developed that provided improved fracture toughness compared with 7075. In the 1980s a new generation of low density Al-Li alloys, e.g., 2090, 8090 and 2091, was developed that offered alternatives, other than increasing strength, for reducing structural weight. During the past decade new improvements have evolved to address the alloy limitations found in pre-1980s aircraft and if used in retrofitting will result in maintenance schedules similar to that required for new aircraft.

The purpose of this paper is to review some recent advances in derivative alloys that have occurred primarily through the use of a very large scientific knowledge base and tighter chemistry and process controls. The newer alloys offer useful improvements in product

performance, quality and reliability and can be applied to aging aircraft problems to dramatically reduce maintenance costs.

2. RECENT ADVANCES IN DERIVATIVE ALLOYS AND TEMPERS

2.1 Improvements in Strength, Corrosion Resistance and Toughness

During the retrofitting of aging aircraft the substitution of alloys with equivalent strength but with higher corrosion resistance and fracture toughness will extend maintenance schedules, decrease down time, and reduce costs. As mentioned previously, RRA offered promise to achieve this goal. In the 1980s work by Wallace and co-workers[5-7] showed that beneficial retrogression and re-aging (RRA) effects can be obtained in large components if the retrogression temperatures are below 200° C for 7075. Hepples et al[8] showed that RRA 7150 using commercially viable thermal process routes can provide material with peak strength and high resistance to SCC and exfoliation corrosion. Based on the RRA concept, Alcoa developed the T77 temper for 7XXX alloys, e.g. 7150. The improvement in the increase in combination of strength/corrosion resistance via the T77 temper process is illustrated in Figure 1.

Alloy 7150-T77 has higher strength with durability and damage tolerance characteristics matching or exceeding those of 7050-T76. Boeing selected extrusions of 7150-T77 as fuselage stringers for the upper and lower lobes of the 777 jetliner because of the superior combination of strength, corrosion and SCC characteristics and fracture toughness. Alloy 7150-T77 plate and extrusions are also being used on the new C17 cargo transport.[9] Improved fracture toughness of 7150-T77 products is attributed to the controlled volume fraction of coarse intermetallic particles and unrecrystallized grain structure, while the combination of strength and corrosion characteristics is attributed to the size and spatial distribution and the copper content of the strengthening precipitates.[9] The improvement in properties using the new temper, relative to older alloys and tempers, is illustrated in Figure 2.

Alloy 7055 was developed by Alcoa for compressively loaded structures.[10] Alloy 7055-T77 plate and extrusions offer a strength increase of about 10% relative to that of 7150-T6 (almost 30% higher than that of 7075-T76). These products provide a high resistance to exfoliation corrosion similar to that of 7075-T76 with fracture toughness and resistance to the growth of fatigue cracks similar to that of 7150-T6. In contrast to the usual loss in toughness of 7XXX products at low temperatures, fracture toughness of 7055-T77 at –65°F (22K) is similar to that at room temperature. Resistance to SCC is intermediate to those of 7075-T6 and 7150-T77 products. A comparison of properties of these 7XXX alloys is given in Table 1. The attractive combination of properties of 7055-T77 is attributed to its high ratio of Zn:Mg and Cu:Mg. When aged to T77 this composition provides a microstructure at and near grain boundaries that is resistant to intergranular fracture and to intergranular corrosion. The matrix microstructure is resistive to strain localization while producing a high strength. Alloy 7055-T7751 is used for the skin of the upper wing surface of the Boeing 777. The improved strength-toughness properties of newer alloys and tempers, relative to the older ones, are illustrated in Figure 3.

2.2 Improvements in Damage Tolerance and Multiple Site Damage

The service life of an airframe can potentially introduce multi-site damage (MSD) states such as widespread fatigue or widespread corrosion that may imperil the structural integrity of the aircraft. For this case, the inspection intervals set by standard residual strength and damage tolerant design that are normally directed at the presence of a single crack, are inadequate. This realization and the desire for reliable. longer lasting aircraft with lower maintenance costs has given rise to requirements that non-pristine or aging structure be accounted for in maintenance strategies.[11] This philosophical shift creates the opportunity for affordable, replacement materials that can not only resist the occurrence of multi-site damage. but which offer improved structural damage tolerance when MSD is present.

The occurrence of widespread damage sites can be associated with the intrinsic characteristics of the material microstructure.[12] Material microstructural sites prone to the development of crack-like damage, attributable to corrosion or fatigue, can be associated with particles, inclusions, pores, and grain boundaries.[13] While these features are necessarily a part of the material, the character of these features can be altered through composition and process modifications while still meeting the required material strength performance characteristics.[14]

Machined structures from plate thicker than three inches is often used to reduce part count and assembly costs associated with built-up components manufactured from thinner material. However, since the thicker plate undergoes less work than thin products there is a higher probability that porosity developed during the casting operation will not be sealed. Obviously, the high porosity material has a poorer fatigue performance than low porosity material. There has been continuous process refinement in the production of thick plate since the early 1980s that has reduced porosity as well as particle and inclusion size. Consequently, the fatigue lifetime of products produced from the more recent material, even for a one-to-one substitution, should be longer than products produced from pre-1980 material. The effect of the process refinement on the fatigue lifetime of 7050-T7451 is illustrated in Figure 4.

Alloy 2024-T3 sheet is often selected for wing and fuselage skins for its superior damage tolerance properties when compared to higher strength 7XXX products. A derivative of 2024-T3, 2524-T3, was recently developed by Alcoa[15] and offers improvement in strength/fracture toughness (approximately 15-20%) and fatigue crack growth resistance (2X) over 2024-T3, Table 2. The improvement was achieved through very tight controls on composition and processing based on the knowledge that constituents associated with Fe and Si impurities lower fracture toughness[16-19] and have an adverse effect on both fatigue crack initiation[13] and fatigue crack growth resistance.[20] Coarse primary phases formed when solubility limits are exceeded at the solution heat treatment temperature (or those formed during hot rolling and not re-dissolved during subsequent processing) have a similar effect.[21] Consequently, tight controls on chemistry, i.e. low levels of Fe and Si, balancing the Cu and Mg content to produce maximum strength without exceeding solubility limits at the solution heat treatment temperature,[22] and a controlled processing schedule are all necessary.[21] In controlling the Cu and Mg contents, the levels of Fe, Si

and Mn in the alloy have to be considered since the constituent phases in 2X24 are usually Al_7Cu_2Fe, $Al_{12}(Fe,Mn)_3Si$, $Al_6(Fe,Cu)$ and the dispersoid is $Al_{20}Cu_2Mn$. The effect of fewer and smaller constituent particles on fatigue initiating corrosion pits is illustrated in Figure 5.

The fatigue crack growth advantage that 2524 has over 2024 enables an increase in operating stress, which offers a weight saving opportunity that may also accommodate mission changes that have occurred in older aircraft. This improvement also allows for an increase in inspection interval, which translates to lower operating costs. Inspections are easier since larger crack sizes can be tolerated and longer critical crack lengths translate to an increase in safety. The effect of skin alloy and operating stress on inspectable crack growth life is illustrated in Figure 6 for a longitudinal fuselage skin crack under an intact frame. Also, 2524 body skin offers substantial residual strength and cyclic life improvements over 2024 in multi-site damage scenarios, Figure 7. The fatigue advantage of 2524 over that of 2024 caries over to corroded material as illustrated in Figure 8. The higher toughness and greater resistance to fatigue crack growth of 2524 resulted in the elimination of tear straps in a weight-efficient manner on the Boeing 777.

2.3 Reductions in Density and Improved Fatigue Crack Growth Resistance

The second generation of Al-Li alloys (the first being the Alcoa alloy 2020) were developed in the 1970s (alloy 1420 in Russia) and the 1980s (alloys 2090, 2091, and 8090). The Al-Mg-Li alloy 1420 and the Al-Li-Cu-X alloys 2090 and 8090 are now in service in the MIG 29, the EH1 helicopter and the C17 transport. Alloy 1420 has only moderate strength and the Al-Li-Cu alloys (which contain approximately 2% lithium) have a number of technical problems, which include excessive anisotropy of mechanical properties, crack deviations, a low stress-corrosion threshold and less than desirable ductility and fracture toughness. Newer Al-Li alloys have been developed that have lower lithium concentrations than 8090, 2090 and 2091. These alloys do not appear to suffer from the same technical problems. The first of the newer generation was Weldalite 049® (2094) which can attain a yield strength as high as 700 MPa and an associated elongation of 10%. A refinement of the original alloy, 2195 which has a lower copper content, is now being used for the U.S. Space Shuttle Super-Light-Weight Tank. Alloy 2195 replaced 2219 and, along with a new structural design, saved 7,500 pounds on the 60,000-pound tank. This allows an increased payload for the Shuttle and reduces the number of flights necessary for the construction of the International Space Station, thus saving millions of dollars.

Three other recent derivatives of the third generation of Al-Li alloys are 2096, 2097 and 2197. They contain lower copper and slightly higher lithium content compared to 2024. Alloys 2097 and 2197 contain a very low Mg content to improve SCC resistance and Mn to prevent strain localization normally associated with the shearable Al_3Li present in the higher Li-containing alloys. Alloy 2097/2197 was recently selected[2] for replacing 2124, which had fatigue problems, for bulkheads on the F16. Alloy 2097 has a 5% density advantage over 2124 and at least 3 times better spectrum fatigue behavior or approximately 15% higher spectrum fatigue stress allowable. Although Al-Li alloys are more expensive than conventional aluminum alloys, the replacement of 2124 by 2097 for

the BL 19 Longeron of the F16 doubles the service life of the part, saving over twenty-one million dollars for the fleet of 850 USAF aircraft.[2] Engine access cover stiffeners, currently made from 2124, are also being replaced by Al-Li alloys due to their better fatigue life. This is an excellent example of retrofitting with improved materials for reducing life-cycle costs as described by Austin et al.[2]

3. SUMMARY AND CONCLUSIONS

Older aircraft can be retrofitted with new materials providing improved DADT when compared to the materials used during the original manufacture of older aircraft. A few scenarios for exploiting the potential benefits of new material replacements are given in Table 3. Continuous improved and derivative variants of existing alloys have the broadest utilization potential. Many of these materials are already flying on new aircraft, e.g., the Boeing 777 and/or have been used for retrofitting aircraft e.g. the F-16. Some alloys may be considered as "preferred equivalents" to their predecessors regardless of application, e.g., 2524 for 2024, and others may be considered "preferred replacements" within limits, e.g. 7XXX-T7X for 7075-T6. However, in order to facilitate retrofitting of aging aircraft with new materials, a generic material substitution system is needed for rapid/broad implementation of the *best* material solutions. This system should include ways to improve the efficiency of the substitution process by substantiating new materials as "preferred replacements," by approving the alloy substitution matrix, and by defining opportunities and cost/benefit trades for replacement scenarios. In addition, the repair and maintenance centers should stock qualified substitutes in order to reduce down time for retrofitting.

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Table 1. Longitudinal Property Comparison in One Inch 7XXX Plate

Alloy Temper	UTS (ksi)	TYS (ksi)	CYS (ksi)	El (%)	Kic ksi(in)$^{1/2}$	ASTM Exco rating	SCC ASTM G47@ 20 days (ksi)
7075-T651	76	69	66	6	20*	ED*	10*
7150-T7751	84	78	77	8	22	EB	25
7055-T7751	89	86	85	7	21	EC	15*

* Typical

Table 2. Typical Mechanical Properties for 2524-T3 and 2024-T3 Sheet in the Long-transverse Direction.

Alloy	Thickness (mm)	UTS (MPa)	TYS (MPa)	Elong (%)	Kc[a] MPa-m$^{1/2}$	da/dN@ Δ K=33[b] (mm/cycle)
2524-T3	0.81 - 1.59	420	303	19	174	2×10^{-3}
	1.60 - 3.26	441	310	21		
	3.27 - 6.32	441	303	22		
2024-T3	0.81 - 1.59	427	296	18	141	6.9×10^{-3}
	1.60 - 3.26	448	310	19		
	3.27 - 6.32	448	310	19		

a) M(T) specimen, T-L orientation, W = 40.6 cm (16 inch), $2a_0$ = 10.2 cm (4 inch) tested per ASTM B 646.

b) T-L orientation, tested per ASTM E 647 under constant Δ K conditions, R = 0.1, relative humidity > 90%.

Table 3. Possible Scenarios for Exploiting the Potential Benefits of New Materials

Repair Option	Primary Requirement	Potential Benefits	Potential Disadvantages	Time/Risks Resources
Identical component/ material replacement	Maintain safety & get it flying	Straightforward	"Prolongs the agony" with high repeat repair costs	Lowest
Form-fit function (material upgrade)	Reduce cost of maintenance, improve readiness	Some capture of new materials benefits	Requires M&P, design and analysis expertise	Moderate
Re-optimize with material upgrade	All of the above plus performance	Greater capture of new materials benefits	Requires extensive M&P, design, fabrication, analysis expertise	Moderate to high
Total redesign with new concept	Maximize life-cycle economics & performance	Full capture of best available technology	Requires full OEM capabilities	Highest

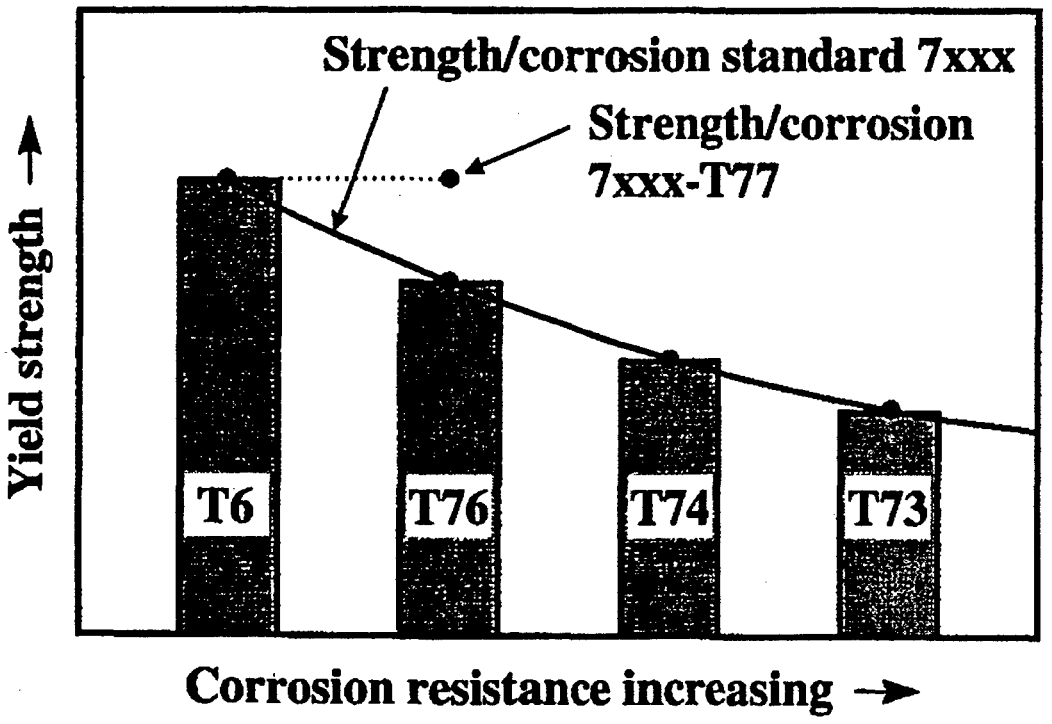

Figure 1. Improvement in strength/corrosion combination due to the T77 temper.

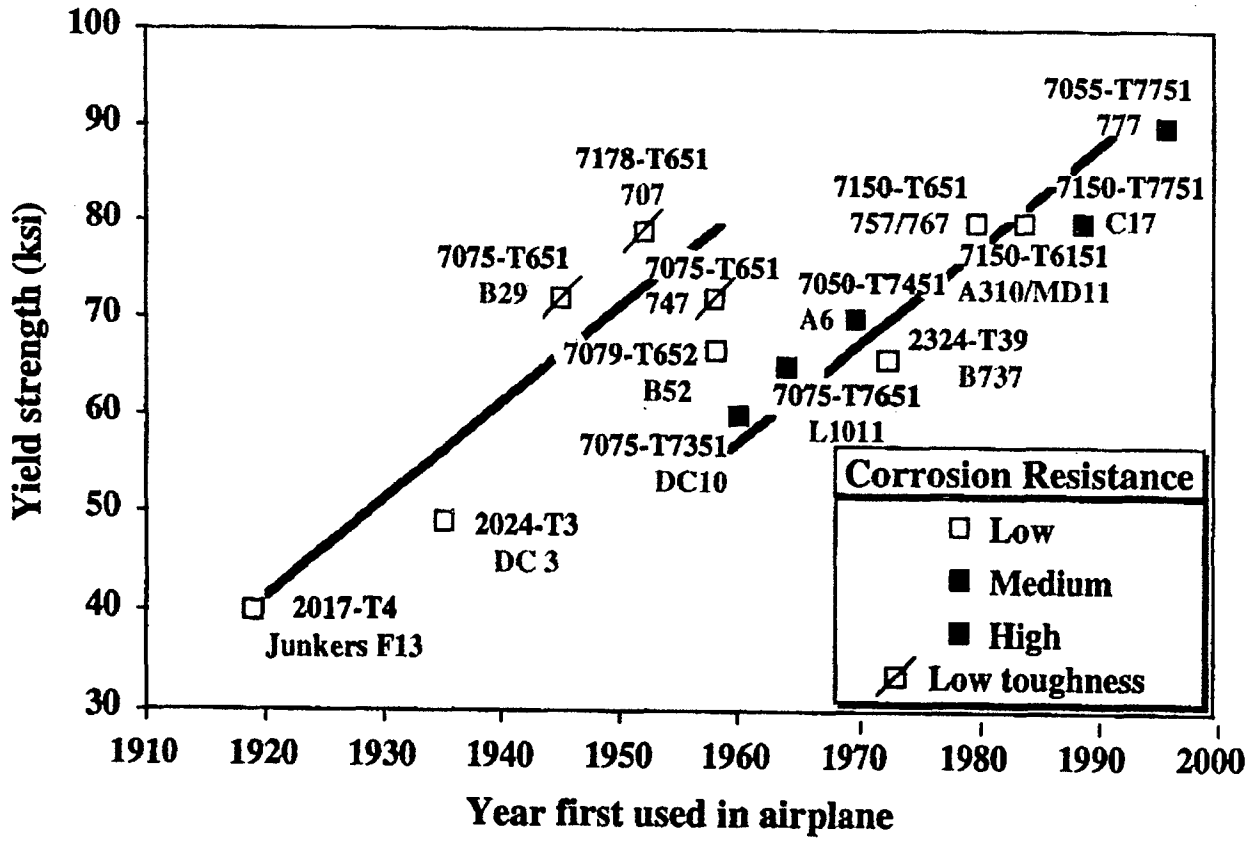

Figure 2. Comparison of strength and corrosion resistance of various aluminum aerospace all as a function of year first used in airplane.

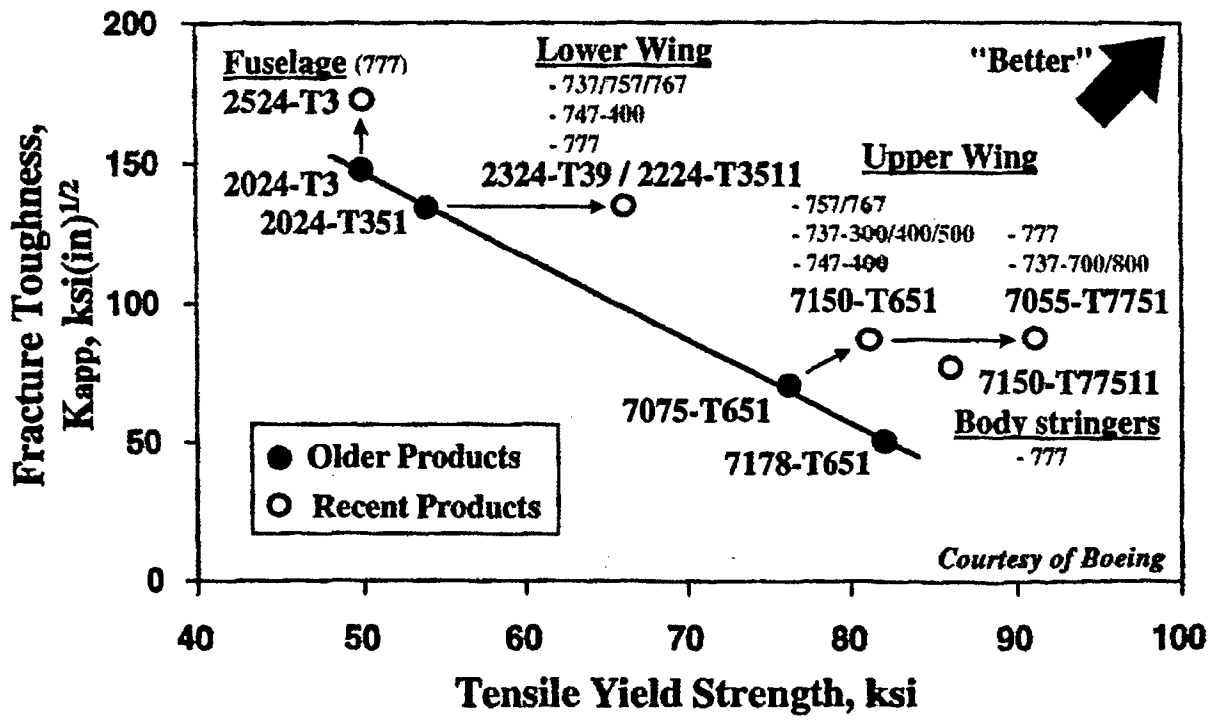

Figure 3. Comparison of fracture toughness/yield strength of older products with newer aluminum products.

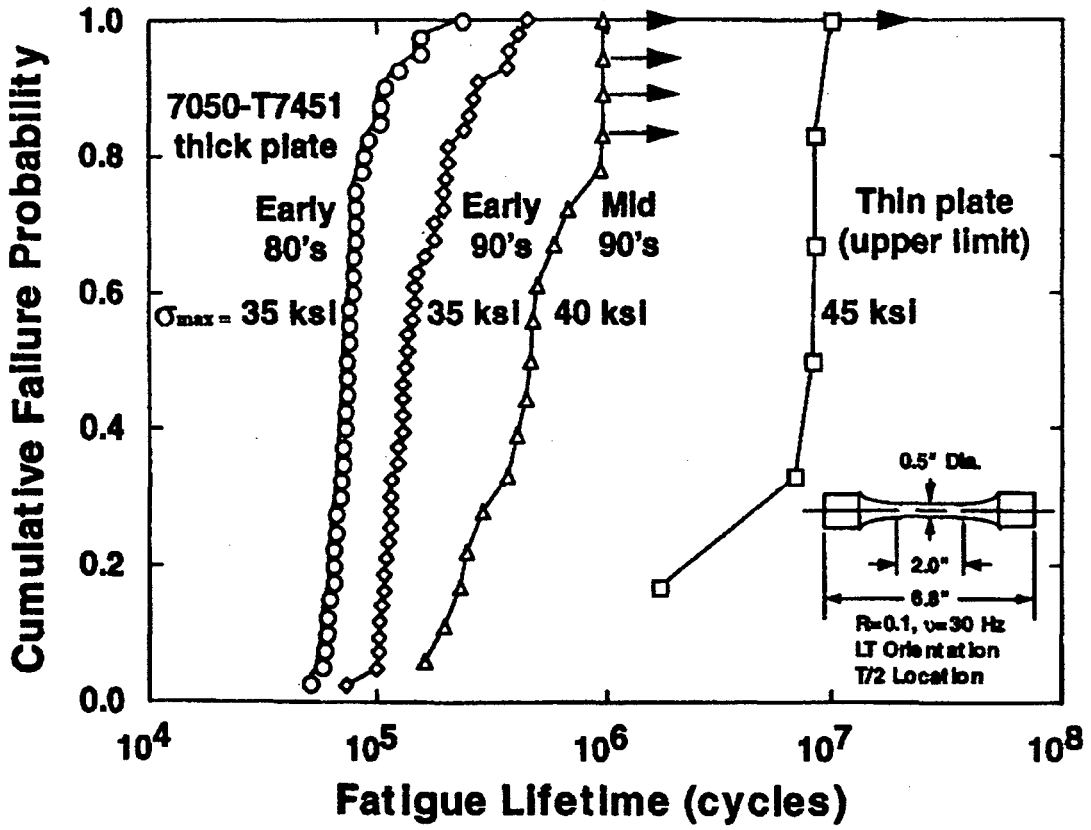

Figure 4. Improvement in fatigue lifetime of 7050-T7451 due to process refinement.

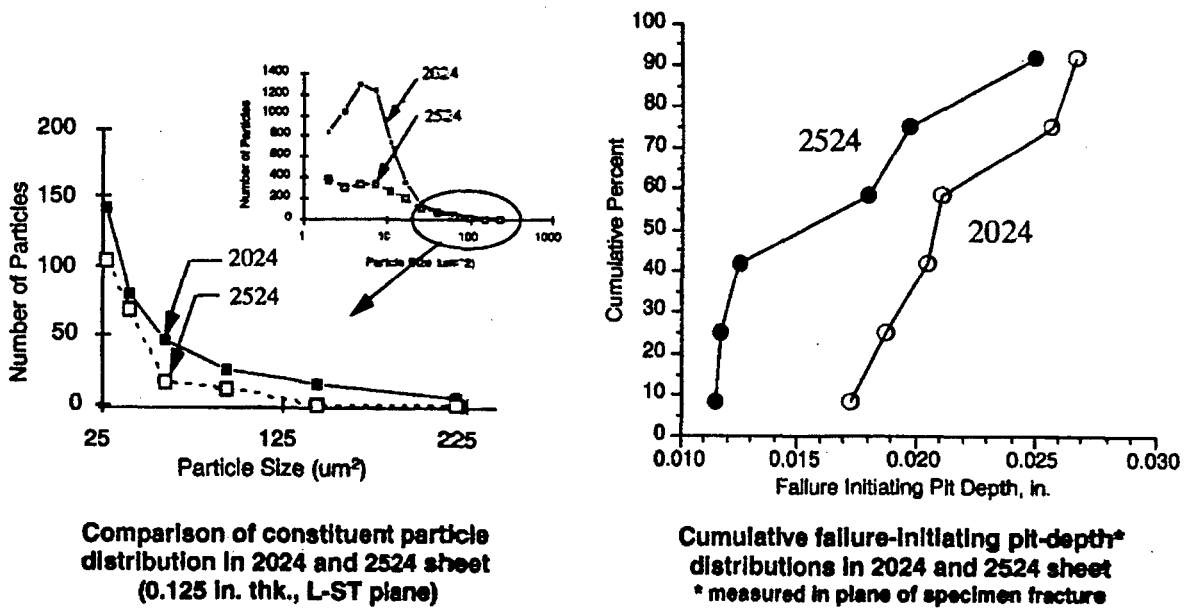

Figure 5. Effect of constituent particle distribution on fatigue-initiating pits.

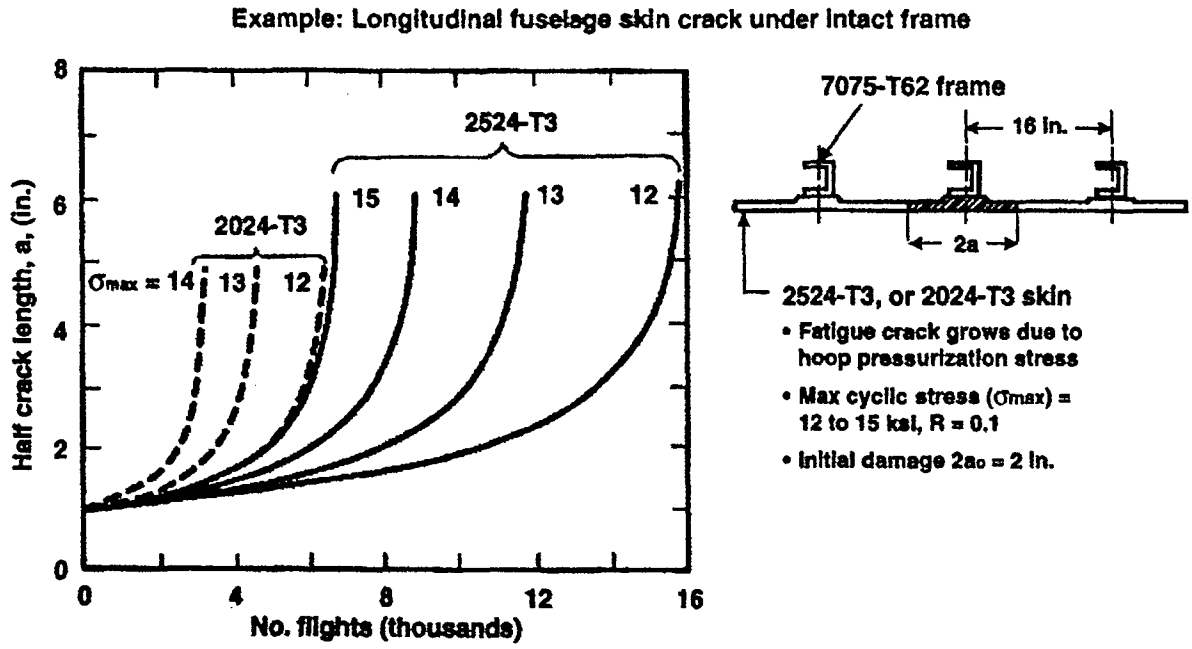

Figure 6. Effect of skin alloy and operating stress on crack growth life.

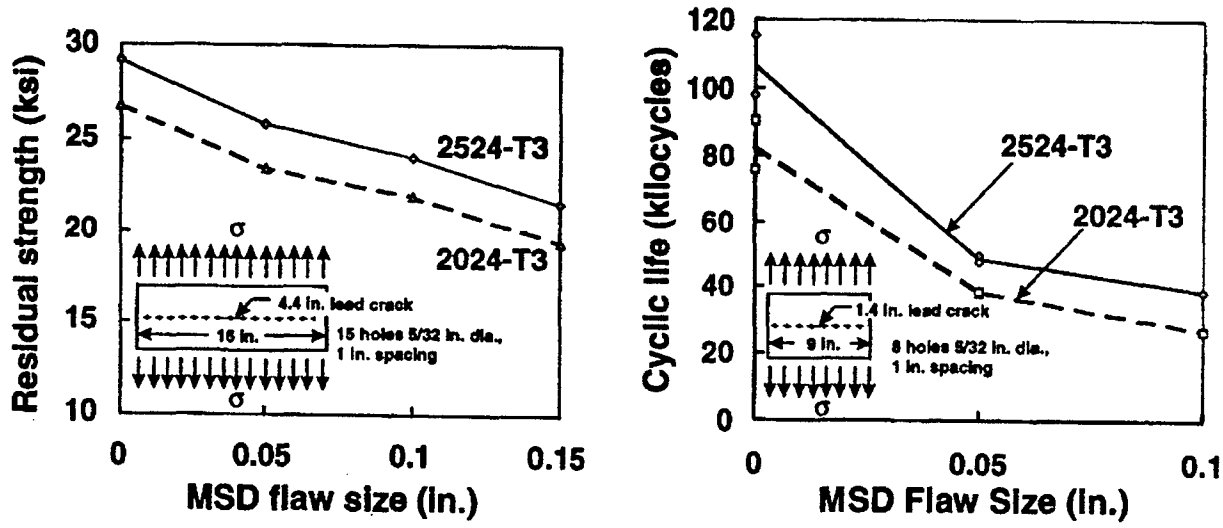

Figure 7. Residual strength and cyclic life capabilities of 2524-T3 and 2024-T3 skin sheet (clad, 0.05 in. thk.) in wide, multi-holed panels with central lead crack and varying size MSD cracks (two per hole).

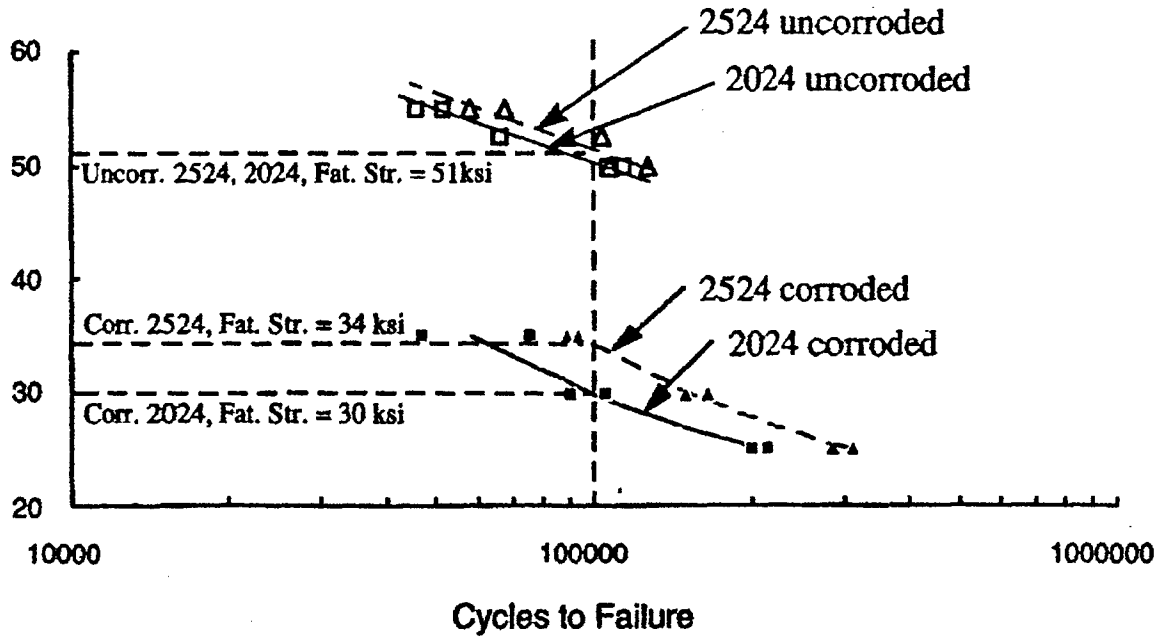

Figure 8. Axial S/N fatigue performance of 2024-T3 and 2524-T3 bare sheet (0.124 in. thk.) with and without prior corrosion.